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HUGHES TOOL COMPANY · AIRCRAFT DIVISION
Culver City, California

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SUPPLEMENTARY STUDY REPORT
ROTOR/WING CONCEPT STUDY,

(11) Feb[redacted]66

(12) 24 p.

(10)

~~Prepared by~~ Robert E. Head

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~~Contract~~ Nonr-4588(00)

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HUGHES TOOL COMPANY - AIRCRAFT DIVISION
Culver City, California

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SUMMARY

An analytical and wind tunnel model test program (Figure 1) was conducted by Hughes Tool Company - Aircraft Division, under contract to the U.S Navy, Office of Naval Research (ONR contract Nonr-4588(00)) and supported jointly by the Bureau of Naval Weapons, to investigate the Rotor/Wing concept. In this concept, the Rotor/Wing is a dual-purpose lifting device for lifting a high-speed VTOL aircraft. It is a high solidity ratio rigid rotor with a central hub fairing large enough to act as a wing and at the same time provide sufficient structural support for the rotor so that it may be stopped in flight. By stopping the rotor in flight, the speed limitations of the helicopter are eliminated. For hovering and low-flight speeds the Rotor/Wing acts as a powered rotor; for cruise flight it is stopped and locked to the fuselage so the large hub may act as a fixed wing with conventional jets propelling the craft.

The major results of the analyses and the wind tunnel tests that were conducted in the David Taylor Model Basin Aerodynamics Laboratory's 8-by-10-foot subsonic wind tunnel were reported in Reference 1. The results of supplementary studies made since publishing that report are presented in this report. These include:

1. Rotor/Wing rolling and pitching moments during the low rpm region of the conversion between stopped- and running-rotor
2. Rotor/Wing shaft bending moments in steady, 1-g, helicopter and autogyro flight
3. Horizontal tail efficiency as a function of the horizontal tail position along the span of the vertical tail
4. Rotor/Wing blade root bending moment comparison between model test data and prediction by IBM 7094 computer

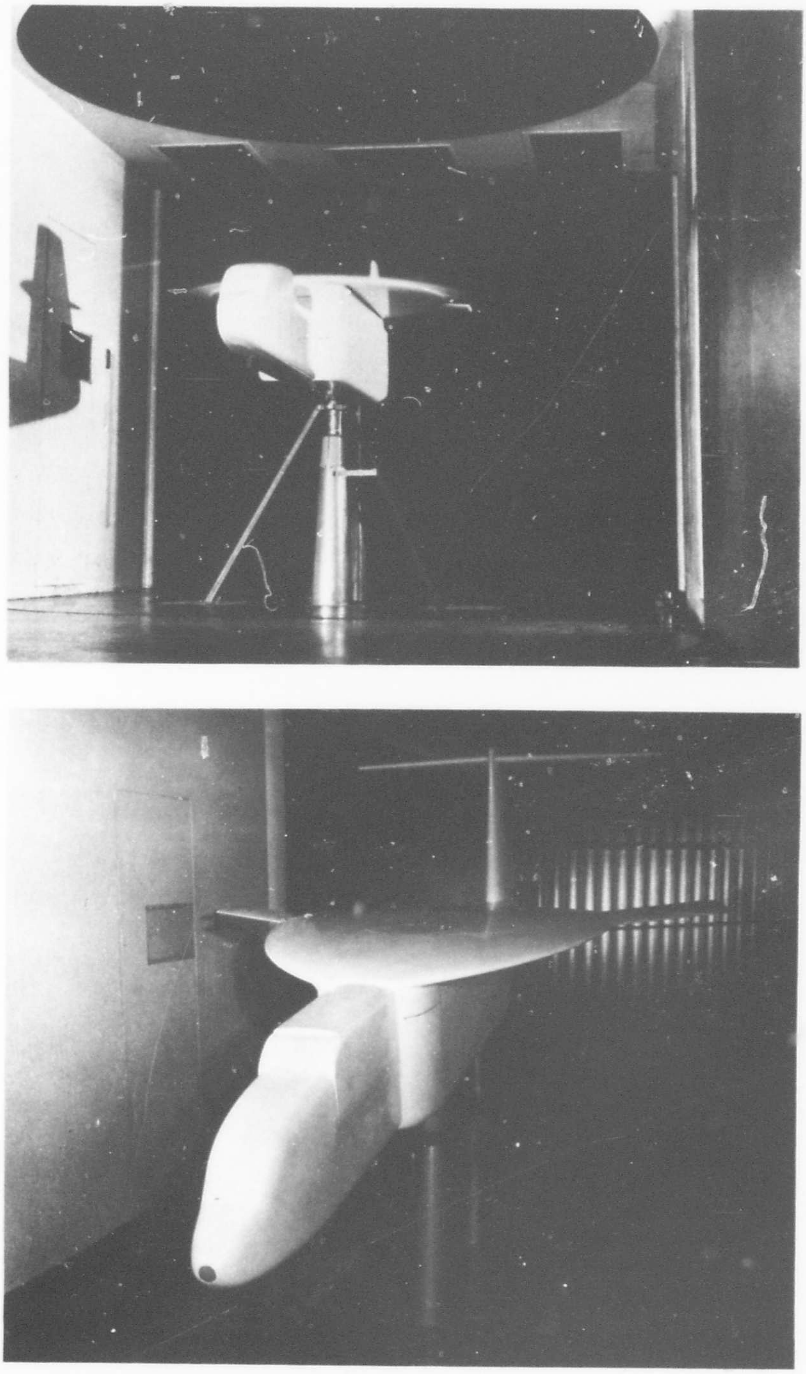


Figure 1. Rotor/Wing Wind Tunnel Model

INTRODUCTION

The Rotor/Wing concept, which was evolved in 1962, consists of a dual-purpose lifting device that is a tip-jet-powered rotor for low-speed flight and stops to become a fixed-wing for high-speed cruising flight. Figure 2 shows schematically the various modes in which the Rotor/Wing vehicle may fly. They may be described as follows:

1. Helicopter

   Where the rotor is powered and control is primarily from rotor cyclic and collective pitch

2. Autogyro

   Where the rotor autorotates and the engines function as conventional turbojets; control is primarily from cyclic pitch augmented by the tail

3. Airplane

   Where the rotor is stopped and locked to act as a fixed wing; the engines function as turbojets; control is from the tail surfaces

The Hot Cycle propulsion system shown schematically in Figure 3 is an integral part of the Rotor/Wing concept. This system transmits power pneumatically from turbojet gas generators to rotor blade tip-jets or airplane-type thrust nozzles; it permits reductions in propulsion system weight and eliminates the need for a torque-reacting tail rotor.

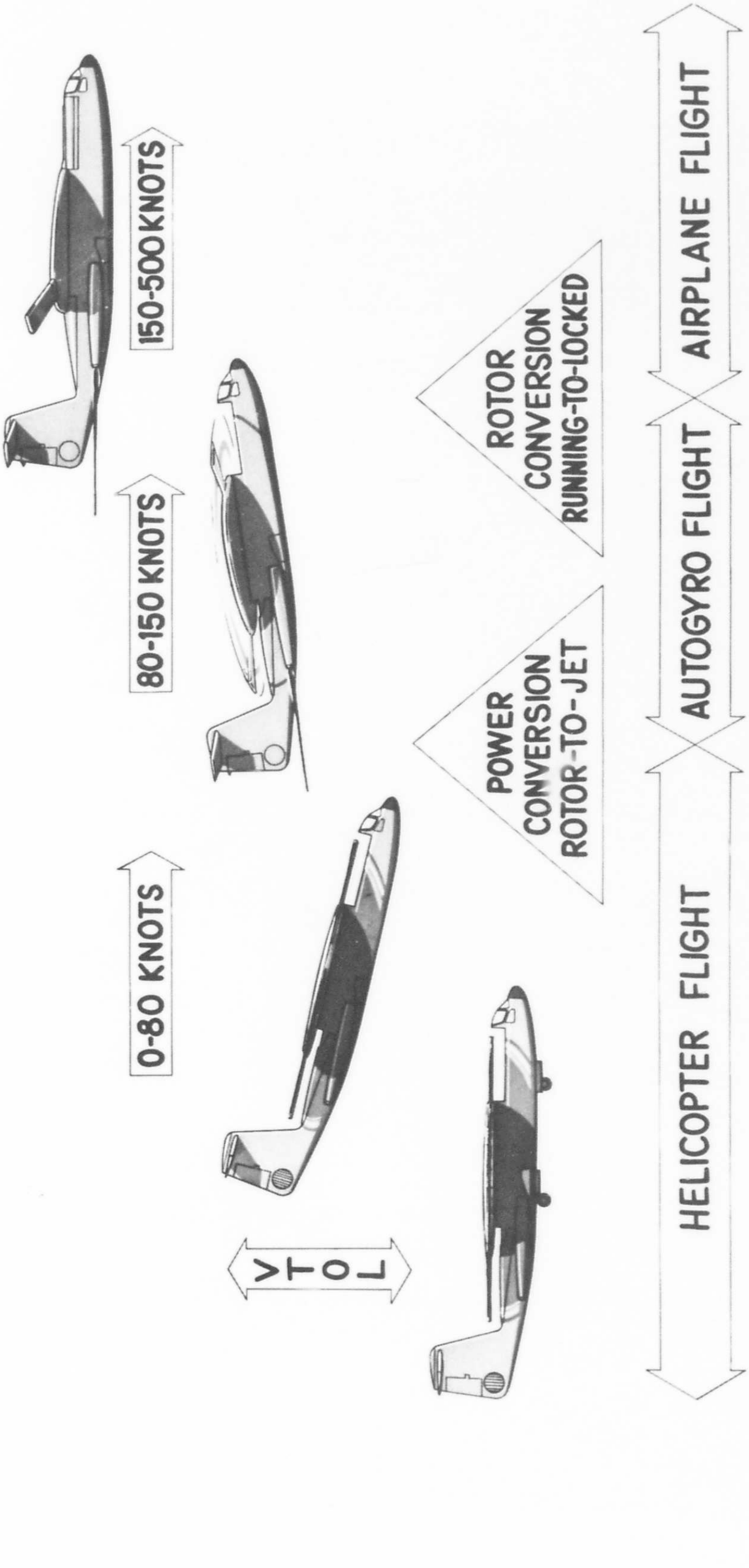


Figure 2. Rotor/Wing Flight Modes

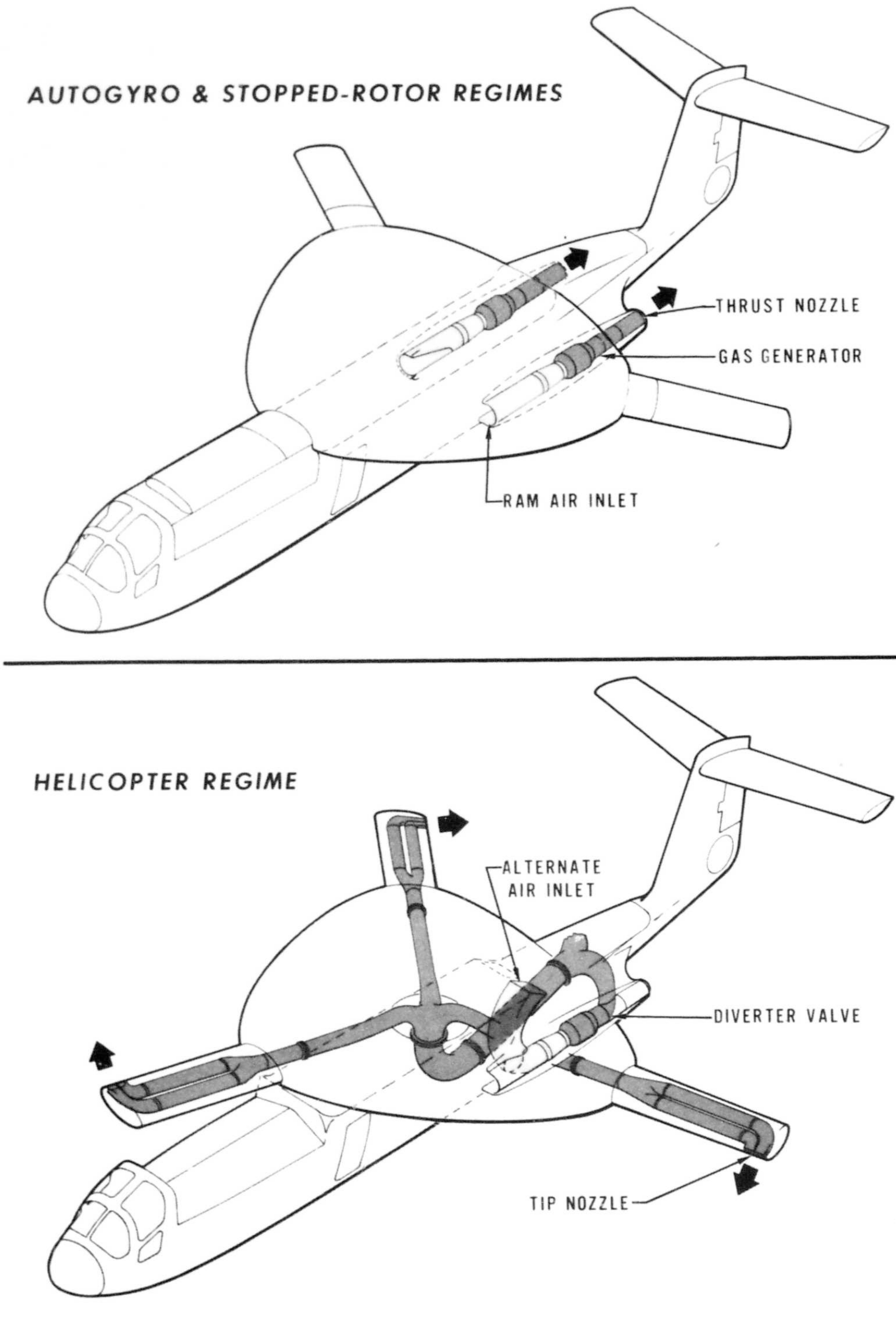


Figure 3. Propulsion System Schematic

The wind tunnel model shown in Figure 1 was tested during two 4-week test periods at the DTMB Aerodynamics Laboratory in the spring of 1965 and the major results of these tests were reported in Reference 1 for four regimes of flight:

Powered rotor

Autorotating rotor

Conversion

Stopped rotor

Since that report was published, supplementary investigation of the test data has been accomplished for certain phases of each of these regimes.

Symbols and nomenclature appropriate to the Rotor/Wing concept study may be found in Table I.

TABLE I
SYMBOLS AND NOMENCLATURE

General

A_1	Longitudinal cyclic pitch control angle, degrees
A_2	2 per revolution longitudinal cyclic pitch angle, degrees
AR	Wing aspect ratio, locked mode
B_1	Lateral cyclic pitch control angle, degrees
D	Drag force, pounds
i_H	Horizontal tail incidence, degrees
Δi_H	Difference in horizontal tail incidence: mean incidence, $i_H = 0°$
L	Lift force, pounds
$\mathcal{L}$	Rolling moment, foot-pounds
M	Pitching moment, foot-pounds
M_H	Blade root bending moment in the chordwise direction, inch-pounds
M_V	Blade root bending moment in the flapwise direction, inch-pounds

$M_{S_\parallel} M_{S_\perp}$	Rotor shaft bending moment in two mutually perpendicular directions, inch-pounds
N	Yawing moment, foot-pounds
N_R	Rotor speed, rpm
Q	Torque, foot-pounds
q	Tunnel dynamic pressure, pound/square foot
R	Rotor radius, feet
S_W	Effective wing area (Rotor/Wing locked), square feet
V	Tunnel airspeed, feet/second
Y	Side force, pounds
α	Fuselage angle of attack, degrees
β	Fuselage side-slip angle, degrees
θ	Rotor collective pitch angle, degrees
θ_1	Blade twist angle, degrees
μ	Rotor advance ratio, $\frac{60\,V}{2\pi N_R R}$
ξ	Ratio of blade root radius to blade tip radius
ψ	Azimuth angle of reference rotor blade, degrees

Configuration

B	Rotor blades
C	Aft hub fairing
F	Fuselage
H	Trisector rotor hub with helicopter-type swashplate, $A_2 = 0°$, Series II
H_1	Trisector rotor hub with cam-type swashplate, $A_2 = 3.5°$, Series III
H_2	Trisector rotor hub with cam-type swashplate, $A_2 = 5°$, Series III
H_3	Trisector rotor hub with cam-type swashplate, $A_2 = 0°$, Series III
I	Image strut system

(inv)	Model inverted
L	Hub locked and sealed to fuselage
N_1	Long faired nose - complete fairing
N_2	N_1 with rotor clearance blocks removed
N_3	Short unfaired nose
P	Powered rotor
S	Horizontal tail (small, Series II)
S_1	Horizontal tail (large, Series III)
V	Vertical tail (small, Series II)
V_1	Vertical tail (large, Series III)
Y	Wing fences
Z	Strut-stiffening braces
Series I	Wind tunnel tests of Rotor/Wing-alone models (June 1964), Reference 2
Series II	Wind tunnel tests of complete model (March 1965); included powered-rotor, autorotating-rotor, pseudoconversion, and stopped-rotor tests; Reference 1
Series III	Wind tunnel tests of complete model (June 1965); included powered-rotor, autorotating-rotor, pseudoconversion, automatic conversion, manual conversion, and stopped-rotor tests; Reference 1

CONVERSION STUDIES

These studies are mainly concerned with extended evaluations of the low-speed end of the conversion, for both Rotor/Wing accelerating and decelerating. Most of the work was done for the accelerating conversion because, unfortunately, the deceleration process was not continued down to zero rpm. The oscillograph data have been converted into time histories of the pitching and rolling moments in the rotor shaft, transferred into the nonrotating fuselage coordinate system. Data are presented for two rotor configurations:

"Conventional" swashplate* with $A_2 = 0°$

Cam-type swashplate* with $A_2 = 5°$

Data for the $A_2 = 5°$ rotor were given in Reference 1, Volume I, Figure 37, wherein the moments were shown to reach a steady-state 3-per-rev character after the first one-half revolution of the rotor and maintain this character as the rpm increased. Figure 4 of the present report contains the same data but carries it out to higher rpm's (the first five revolutions of the rotor are included, bringing the rpm up to 85, which is 14 percent of the helicopter design rotor speed), and shows that the steady-state 3-per-rev character of the curve is maintained throughout this rpm range.

Figure 5 shows similar data for the $A_2 = 0°$ rotor, but here the predominant characteristic is a 1-per-rev moment of a magnitude greater than that of the oscillating moment for the $A_2 = 5°$ rotor, thus again showing the benefit of the cam-type swashplate control for reducing the loads transferred from the

*Pitch control of the rotor blades has the conventional NACA definition:

$$\theta = A_0 - A_1 \sin \psi - B_1 \cos \psi - A_2 \sin 2\psi - \ldots.$$

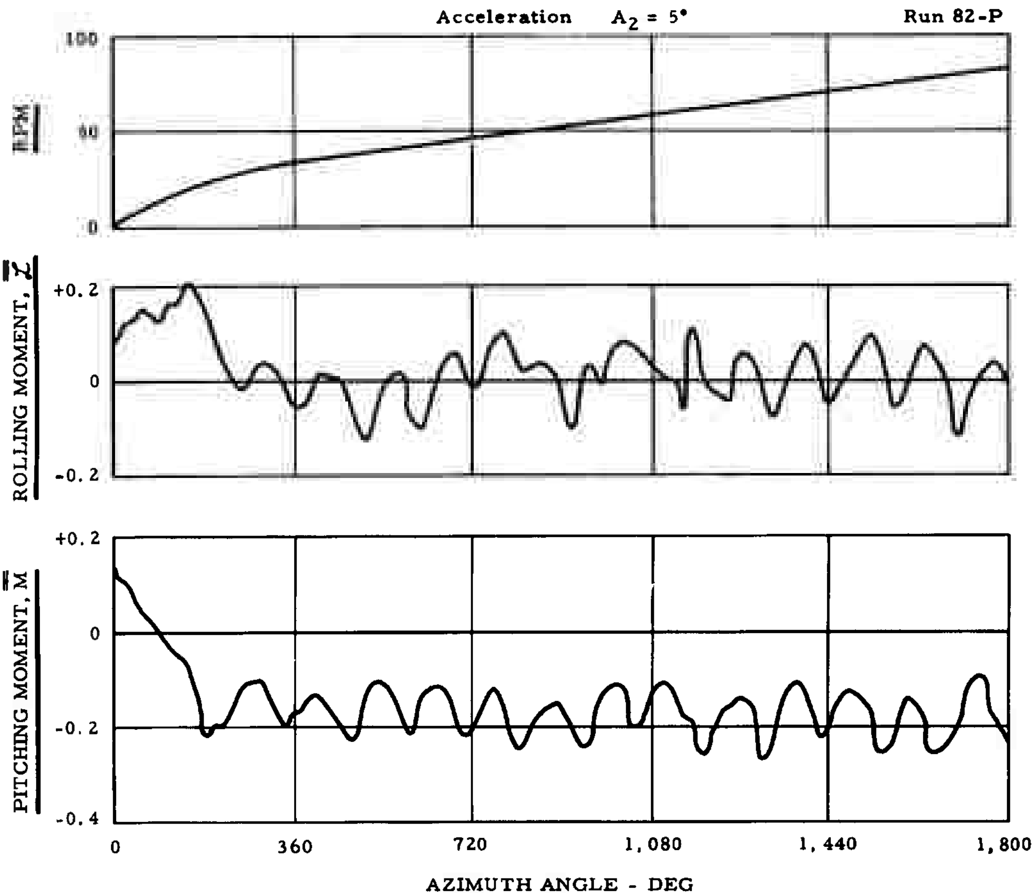


Figure 4. Time History - Rotor Shaft Bending Moments - Rotor Acceleration During Conversion - Low RPM - $A_2 = 5°$

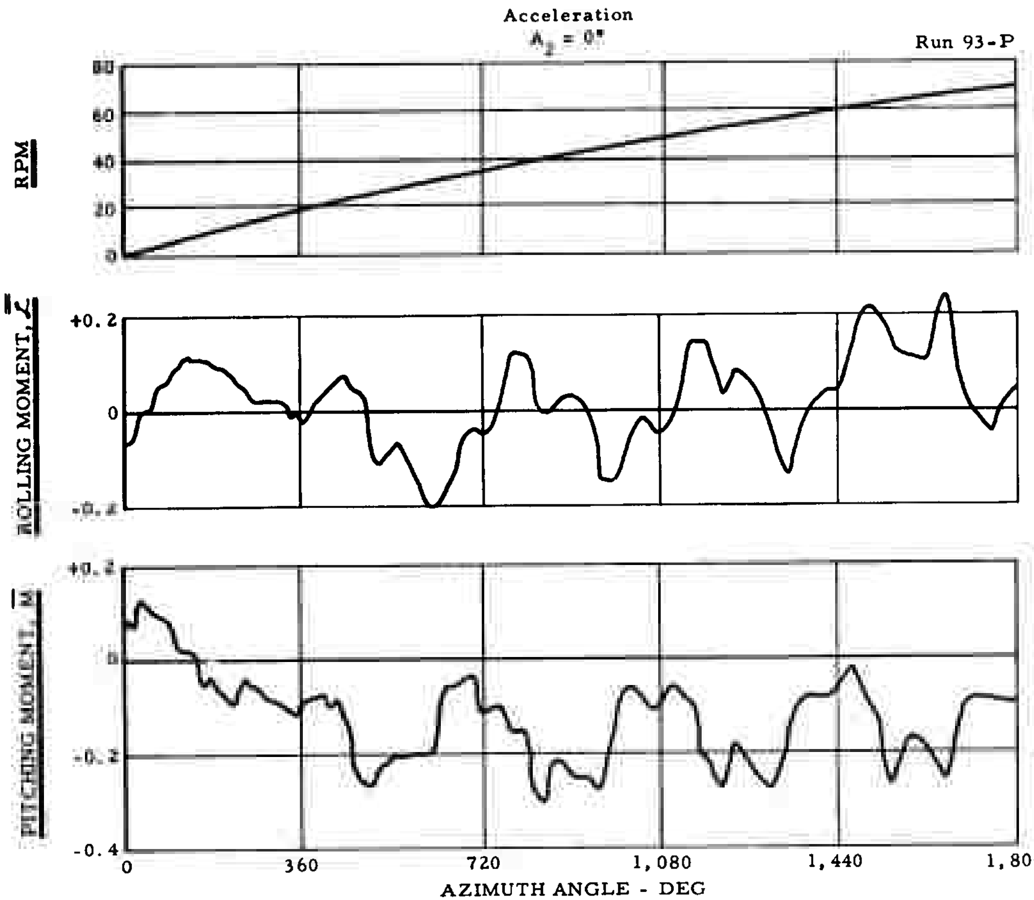


Figure 5. Time History - Rotor Shaft Bending Moments - Rotor Acceleration During Conversion - Low RPM - $A_2 = 0°$

Rotor/Wing into the fuselage. Figure 6 shows that this 1-per-rev trend is carried to higher rpm's.

Figure 7 shows the moments for the $A_2 = 5°$ rotor in deceleration at several rpm's. Here, in deceleration the moment characteristics are similar to those in acceleration; that is, the curves are predominantly 3 per rev and of approximately the same magnitude as in acceleration.

Figure 8 is for deceleration for the $A_2 = 0°$ rotor. The same tendency as in acceleration is noted -- a 1-per-rev moment with greater amplitude than that of the $A_2 = 5°$ rotor.

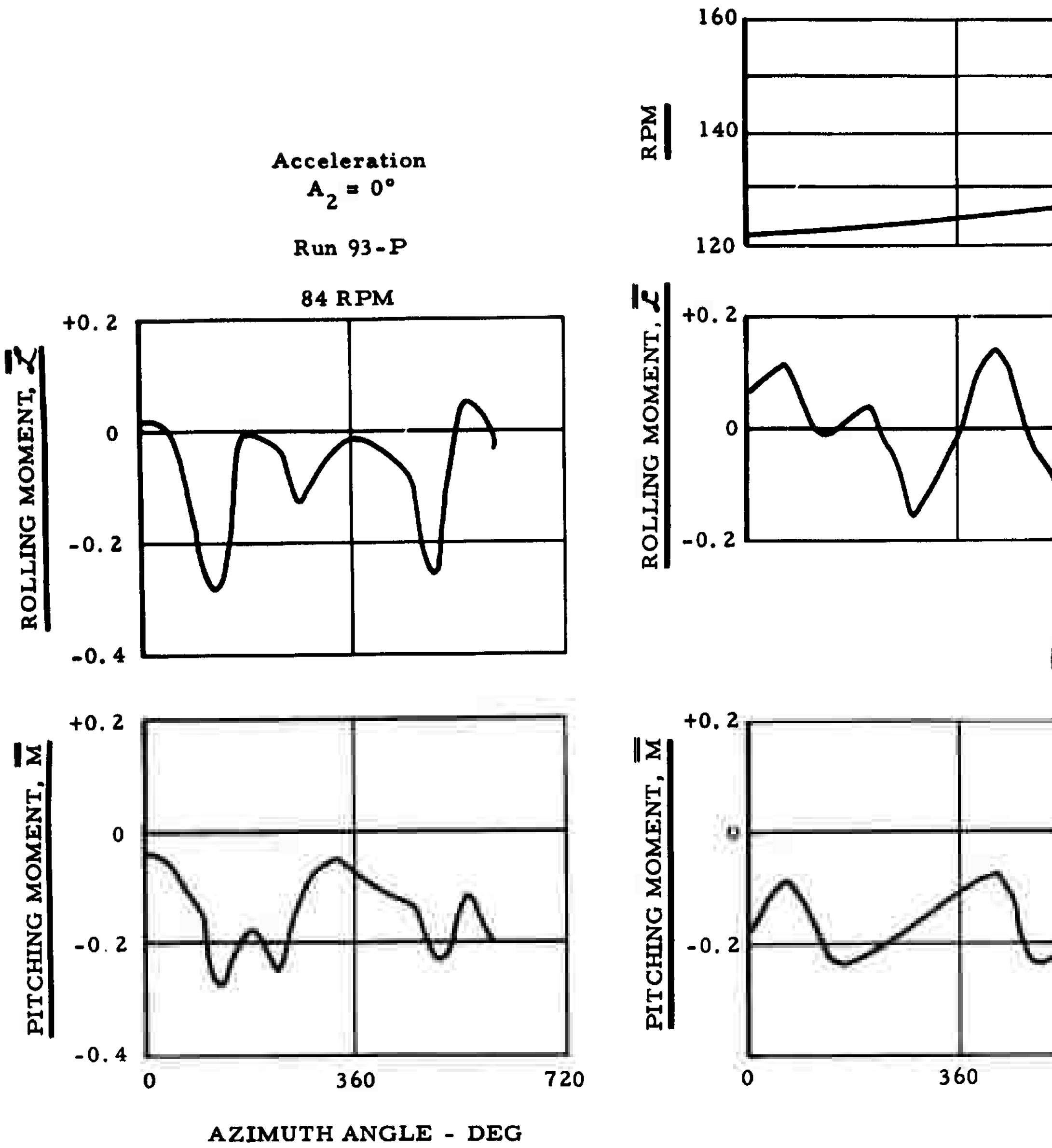
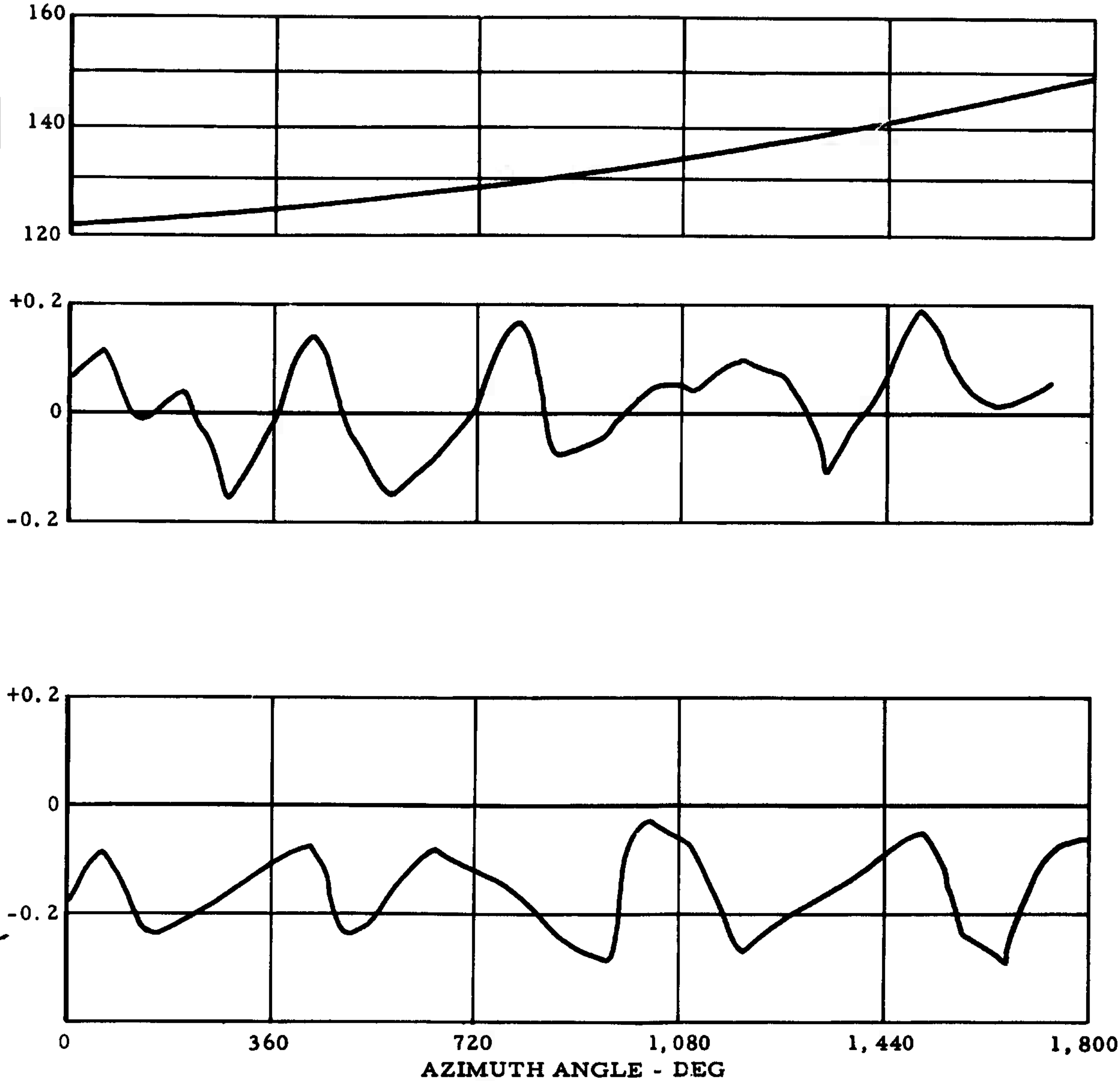


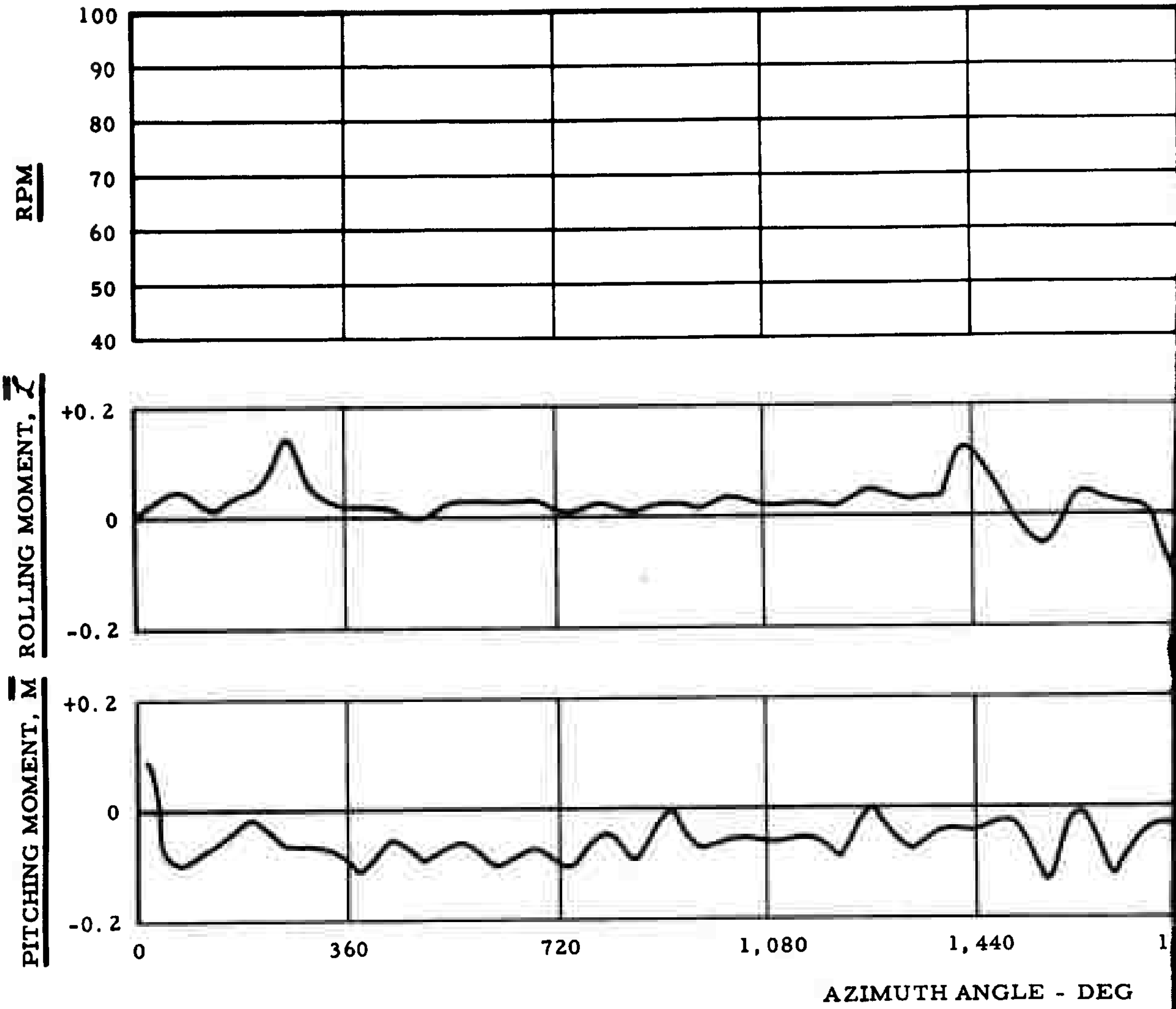
Figure 6. Time History - Rotor Shaft Bending Moment - Rotor Acceleration During Conversic

A



Acceleration During Conversion - Intermediate RPM - $A_2 = 0°$

B


RPM
100
90
80
70
60
50
40
ROLLING MOMENT, $\bar{\mathcal{L}}$
+0.2
0
-0.2
PITCHING MOMENT, $\bar{M}$
+0.2
0
-0.2
0
360
720
1,080
1,440
AZIMUTH ANGLE - DEG

A

Deceleration $A_2 = 5°$ Run 82-P

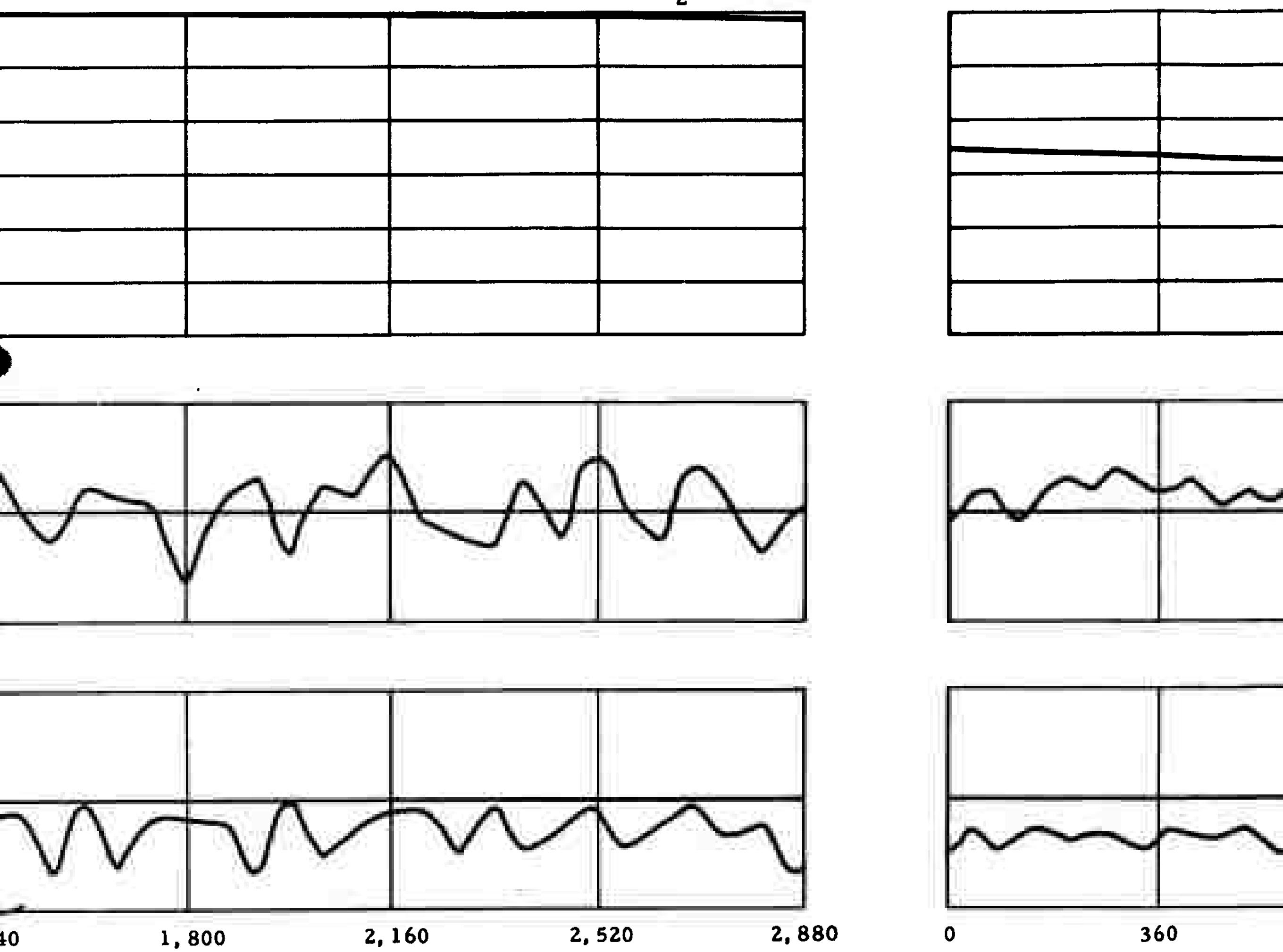
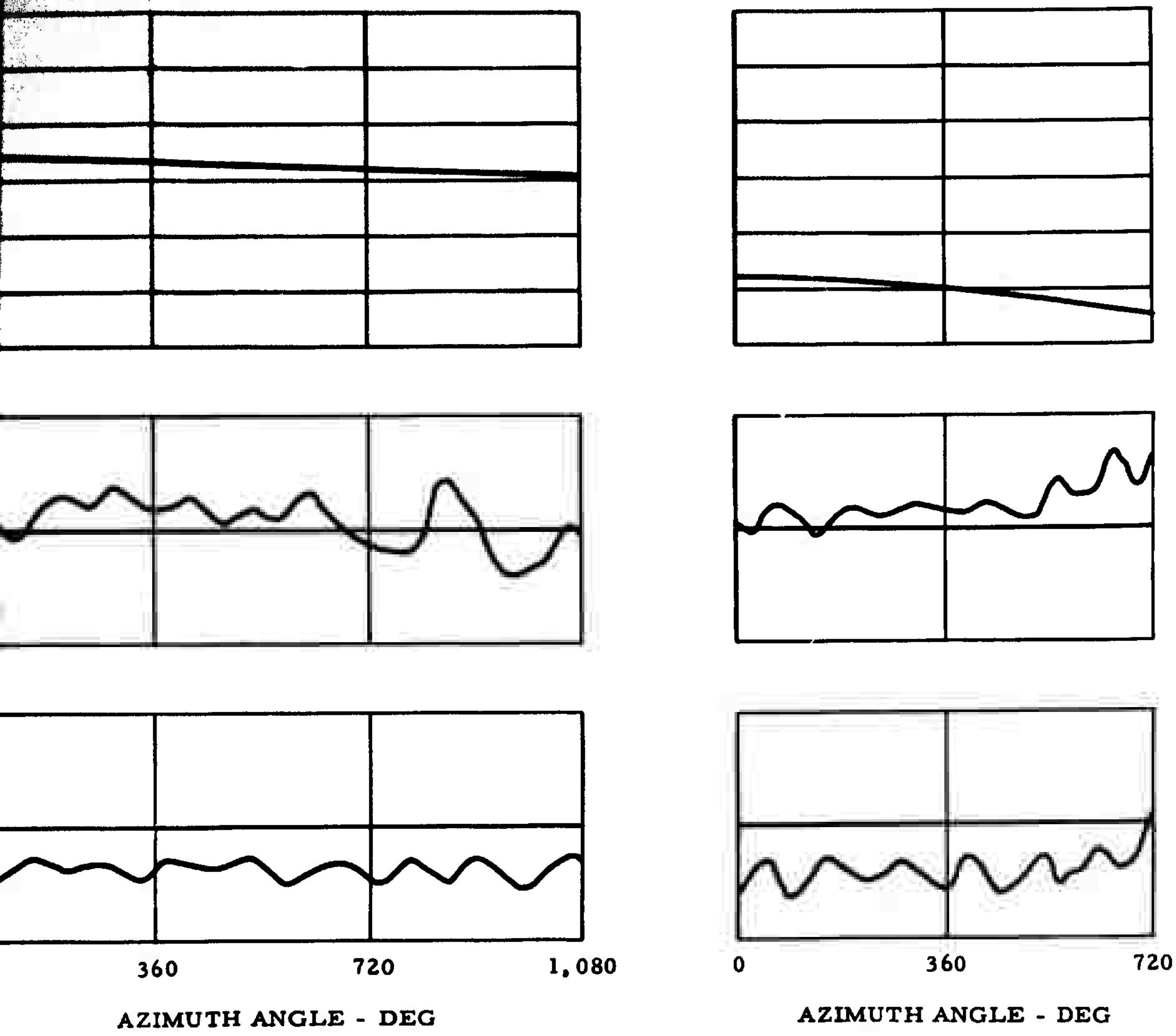


Figure 7. Time History - Rotor Shaft Bending Moment - Rotor Deceleration During Conve

B



celeration During Conversion - Intermediate RPM - $A_2 = 5°$

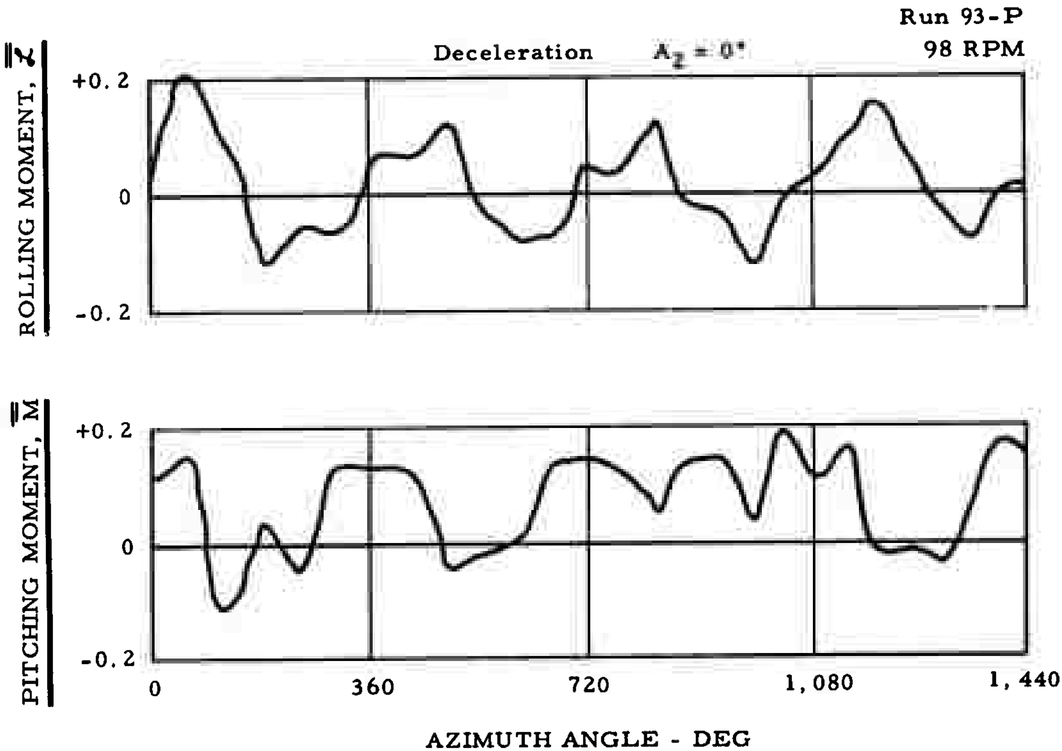


Figure 8. Time History - Rotor Shaft Bending Moment - Rotor Deceleration During Conversion - Intermediate RPM - $A_2 = 0°$

HELICOPTER AND AUTOGYRO STUDIES

Harmonic analyses were made of the Rotor/Wing shaft bending moment oscillograph records to determine the detailed effect of changing the rotor control swashplate from the conventional ($A_2 = 0°$) swashplate to the cam-type swashplate ($A_2 = 5°$). Conditions approximating level flight were chosen for powered rotor flight at advance ratios of 0.25 and 0.35, and for autorotation at an advance ratio of 0.35. The bending moments were nondimensionalized by dividing by rotor lift and rotor radius, and are listed in Table II.

These moments were measured in the rotating coordinate system of the rotor, so when transferred into the nonrotating fuselage system, the first harmonic moment becomes a steady moment, the second harmonic results in oscillating moments in the fuselage, and the third harmonic causes a vertical bounce in the fuselage.

A comparison between the rotor shaft bending moments for the conventional swashplate ($A_2 = 0°$) and the 2-per-rev swashplate ($A_2 = 5°$) shows that the 2-per-rev cyclic pitch input is a very powerful means for reducing the oscillating components (2-per-rev and 3-per-rev) of the shaft bending -- the reduction being to between one-half and one-third. This is an even more powerful effect than was shown in Table I (page I-46) of Reference 1, Volume I, for reducing the oscillating loads during conversion.

TABLE II

ROTOR SHAFT BENDING MOMENT HARMONIC ANALYSIS

Flight Mode	Advance Ratio	$M_{S_{\perp}}^{(1)(2)}$						$M_{S_{\parallel}}^{(1)(2)}$					
		$A_2 = 0°$			$A_2 = 5°$			$A_2 = 0°$ (3)			$A_2 = 5°$		
		1st	2nd	3rd	1st	2nd	3rd	1st	2nd	3rd	1st	2nd	3rd
Helicopter	0.25	0.168	0.445	0.086	0.033	0.208	0.060	-	-	-	0.049	0.104	0.016
Helicopter	0.35	0.040	1.162	0.081	0.182	0.482	0.110	-	-	-	0.026	0.296	0.034
Autogyro	0.35	0.920	0.881	0.096	0.465	0.397	0.044	-	-	-	0.391	0.453	0.056

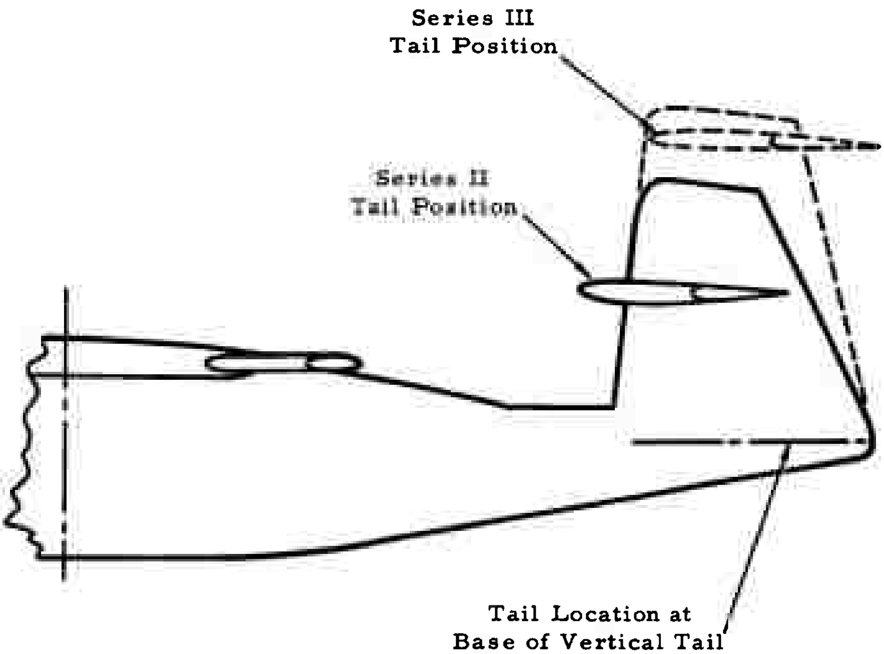
Note:
1. Bending moments are in the coordinate system that rotates with the rotor.
2. Bending moments made nondimensional by dividing by rotor lift and rotor radius.
3. Strain gage circuit for this parameter was inoperative.

HORIZONTAL TAIL EFFECTIVENESS STUDIES

The downwash angle and dynamic pressure ratio data plotted in Figures 58 and 59 of Volume I of Reference 1 were used as the basis for studying horizontal tail effectiveness as a function of position along the vertical tail span. These data were cross-plotted to find the downwash angle and q-ratio distributions along the horizontal tail semispan for the three tail heights indicated in Figure 9. The left-hand portions of Figures 10 and 11 show these distributions for various model angles of attack. The right-hand portions of these figures are plots of the average q-ratio or downwash angle over the semispan of the horizontal tail versus model angle of attack.

Figure 10 shows that the high tail experiences free-stream dynamic pressure to more than 12-degree angle of attack, but then the q-ratio falls off rapidly with increasing α. In the low tail position, the overall level of (q_T/q) is lower but the rate of change at high α's is also less, while in mid position the q-ratio is about average between the two extremes.

Figure 11 shows that for the high tail position the slope of the curve of average downwash angle with angle of attack is approximately 0.35 degree per degree up to approximately α = 15 degrees, and then increases very rapidly at higher angles. The low tail position experiences a $d\epsilon/d\alpha$ value of approximately 0.22 up to approximately 12-degree angle of attack; above this, the slope increases at approximately half the rate of that for the high tail position. At the mid position, the $d\epsilon/d\alpha$ slope is greater than for either of the other two positions.



From Reference 1, Figure 8

Figure 9. Horizontal Tail Locations

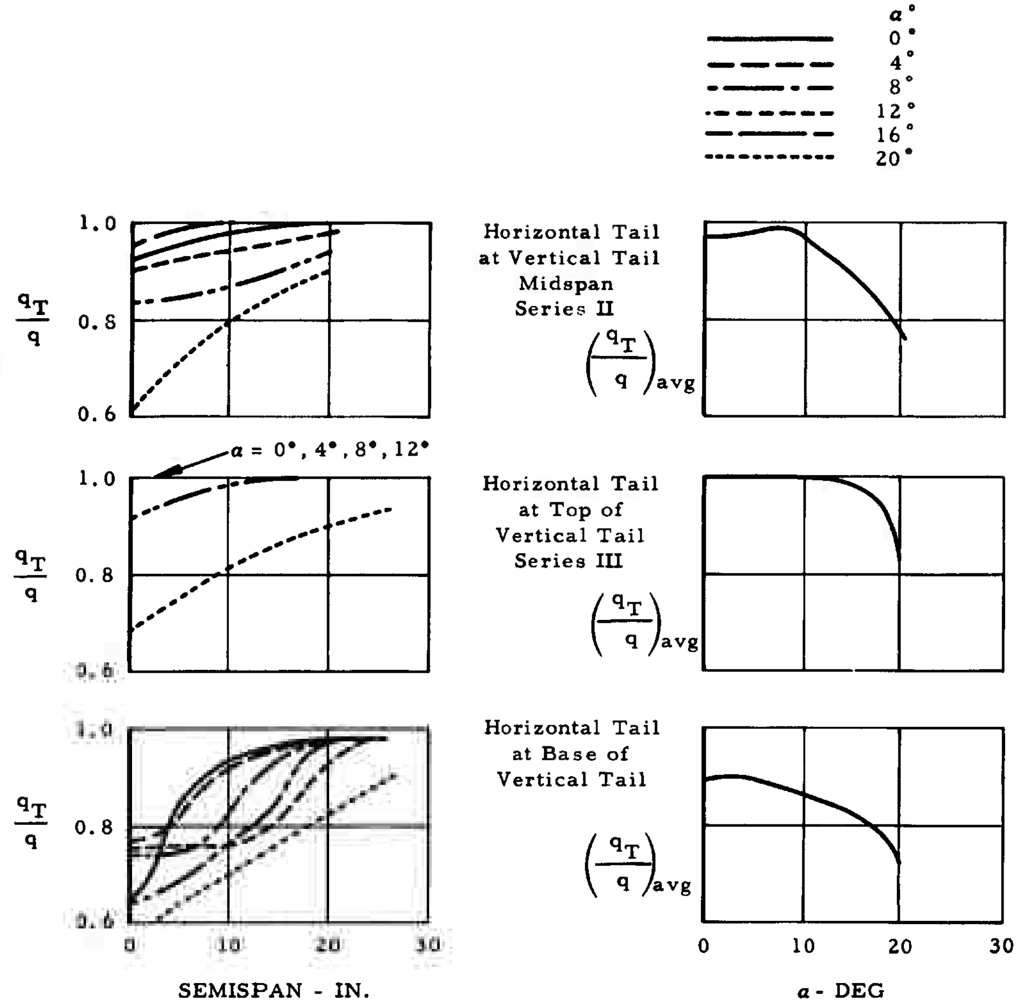


Figure 10. Rotor/Wing q-Ratio at Horizontal Tail

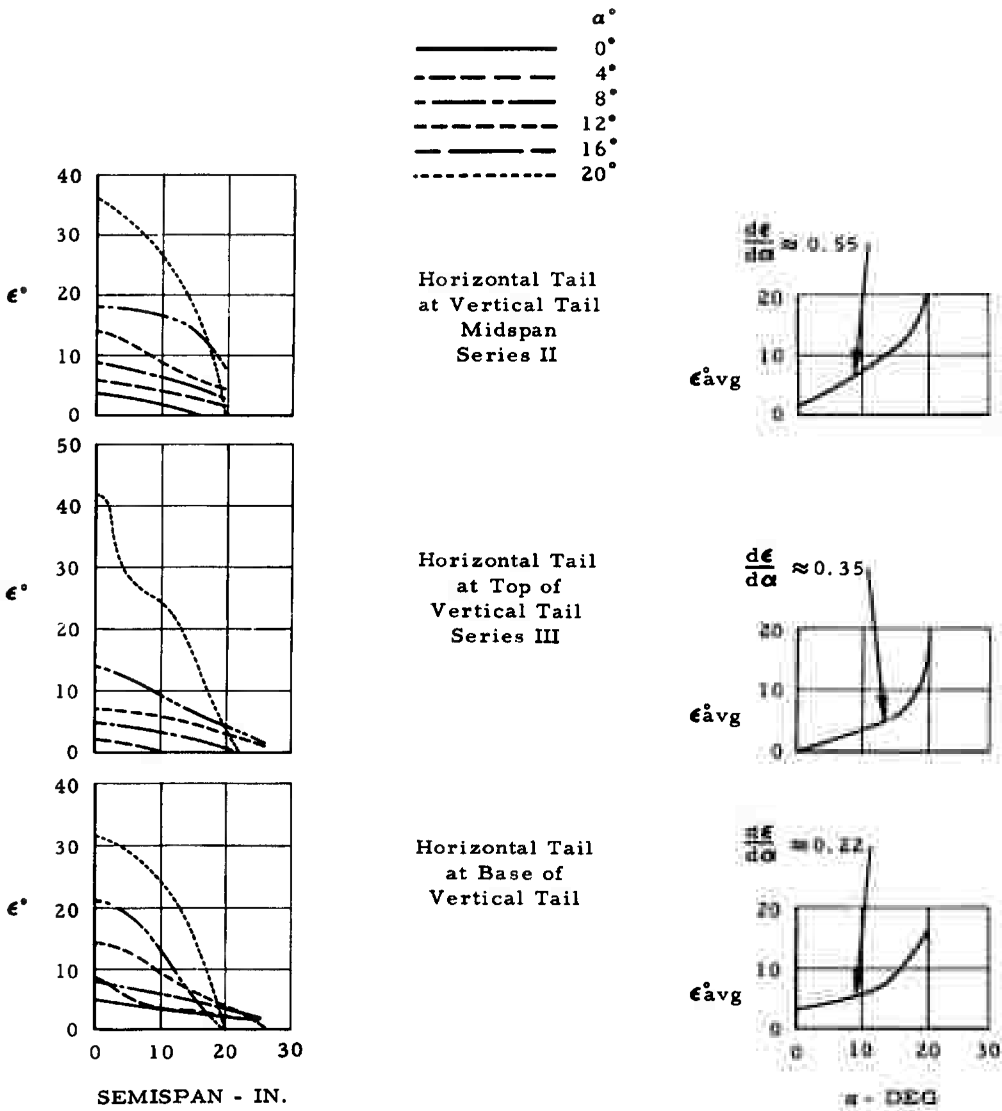


Figure 11. Rotor/Wing Downwash at the Horizontal Tail

Combining these q-ratio and $(d\epsilon/d\alpha)$ values into tail efficiency:

$$\eta = \frac{q_T}{q}\left(1 - \frac{d\epsilon}{d\alpha}\right)$$

$$\eta_{high} \approx 1.00\ (1 - 0.35) \approx 0.65$$

$$\eta_{mid} \approx 0.96\ (1 - 0.55) \approx 0.43$$

$$\eta_{low} \approx 0.87\ (1 - 0.22) \approx 0.68$$

Thus, the tail efficiency is approximately equal for the horizontal tail mounted at either the tip or the base of the vertical tail, but is only approximately two-thirds as effective at the mid position. Because the tail effectiveness is nearly the same for both the high and the low positions over the angle of attack range up to approximately 12 degrees and then the fall-off of effectiveness with increasing angle of attack at higher angles is smaller for the low tail position, the low position is the most appropriate from an aerodynamic standpoint. This low position would also lead to a simpler and lighter weight vertical tail structure than in the cases where the horizontal is located higher on the vertical tail.

ROTOR BLADE ROOT BENDING MOMENT COMPARISON

The Rotor/Wing blade root bending moments were measured by strain gages during the wind tunnel tests at the David Taylor Model Basin Aerodynamics Laboratory and are reported in Reference 1. An existing computer program for helicopter blade loads analysis, used with an IBM 7094 digital computer, was modified to calculate blade root bending moments of the Rotor/Wing for comparison with the model test data. The numerical integration program considered three degrees of freedom for the blades: first blade flapwise bending mode and first and second blade torsion modes.

The model test data used for the comparison between test and computer were taken for a helicopter test point measured during run 42-A of the Series-II wind tunnel tests. The conditions were:

μ = 0.25

q = 3.8 lb per sq ft

RPM = 600

α_{hub} = -5.5 deg

θ = 15 deg

A_1 = 0 deg

B_1 = 7.2 deg

A_2 = 0 deg

L = 55.1 lb

M_V = 102 in-lb

The overall model pitching and rolling moments for this run were compared with those of a similar run (run 91-P) in which the blades were off, to determine

the pitching and rolling moments contributed by the blades themselves. These moments, in addition to the rotor lift and propulsive thrust, determined the proper flight condition of the test. Cyclic pitch values, A_1 and B_1, were varied in the computer study until this flight condition was achieved; then the blade root bending moments were obtained. These computed moments were due only to aerodynamic loads; the inertia relief due to centrifugal force was neglected. Therefore, these moments are expected to be higher than measured.

The proper flight condition was achieved at $B_1 = 10.4°$ and $A_1 = 2.2°$. For this condition, the blade root bending moment was 261 inch-pounds. Because the theory assumed constant inflow, the variation of cyclic pitch and blade root bending moment from the test value ($A_1 = 0°$, $B_1 = 7.2°$, $M_V = 102$ in-lb) is thought to be mainly the result of nonuniform inflow in the model test condition.

The derivatives of pitching and rolling moment with A_1 and B_1 were also computed and compared with the test data presented in Figure F-20 of Reference 1, Volume I. Table III indicates that the computed derivatives compare quite favorably with the test results.

TABLE III
PITCHING AND ROLLING MOMENT DERIVATIVES

Advance Ratio = 0.25

Derivative	Theory	Test
dC_M/dA_1	0.0108/deg	0.0160/deg
$dC_{\mathcal{L}}/dA_1$	0.0012/deg	0.0020/deg
dC_M/dB_1	-0.0073/deg	-0.0060/deg
$dC_{\mathcal{L}}/dB_1$	0.0179/deg	0.0210/deg

CONCLUSIONS AND RECOMMENDATIONS

This supplementary study of the Rotor/Wing concept indicates the following:

1. Second harmonic cyclic pitch input to the rotor controls is a powerful means of reducing rotor blade root bending moments and rotor shaft bending moments in powered rotor flight, steady-state autorotating rotor flight, and conversion flight.

   Because only one value of second harmonic cyclic pitch input was investigated, which may or may not be optimum, further investigation of the effect of this parameter should be carried out.

2. The horizontal tail should be located at either the base or the tip of the vertical tail for best aerodynamic efficiency. Structural efficiency is unquestionably best for the low position of the horizontal tail; hence, a low position for the horizontal tail is recommended.

3. Presently available digital computing techniques result in somewhat conservative estimates of Rotor/Wing blade bending moments. Further refinements in programming are required to obtain closer agreement between test and theory.

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2. Matthews, J. T., Low-Speed Wind Tunnel Investigation of a Circular and a Triangular Rotor/Wing Model in the Stopped-Rotor Mode, Test Report AL-23, David Taylor Model Basin Aerodynamics Laboratory, October 1965.

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13. ABSTRACT

The basic data and analyses from the Rotor/Wing wind tunnel tests were reported in HTC-AD 65-15 (AD 473 862L, 863L, 864L). Supplementary analyses of the data with respect to Rotor/Wing rolling and pitching moments in the low-rpm range, Rotor/Wing shaft bending moments in 1-g running-rotor flight, horizontal tail efficiency, and Rotor/Wing blade root bending moments are included in ~~the present~~ report.

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Wing						
Stopped-Rotor						
Conversion						
Helicopter						
Wind Tunnel Test						

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It is highly desirable that the abstract of classified reports be unclassified. Each paragraph of the abstract shall end with an indication of the military security classification of the information in the paragraph, represented as *(TS)*, *(S)*, *(C)*, or *(U)*.

There is no limitation on the length of the abstract. However, the suggested length is from 150 to 225 words.

14. KEY WORDS: Key words are technically meaningful terms or short phrases that characterize a report and may be used as index entries for cataloging the report. Key words must be selected so that no security classification is required. Identifiers, such as equipment model designation, trade name, military project code name, geographic location, may be used as key words but will be followed by an indication of technical context. The assignment of links, rules, and weights is optional.